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RESEARCH MEMORANDUM

ANALYTICAL INVESTIGATION OF FUEL TEMPERATURES AND FUEL-EVAPORATION LOSSES ENCOUNTERED IN LONG-RANGE HIGH-ALTITUDE SUPERSONIC FLIGHT

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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ANALYTICAL INVESTIGATION OF FUEL TEMPERATURES AND FUEL-EVAPORATION LOSSES ENCOUNTERED IN LONG-RANGE HIGH-ALTITUDE SUPERSONIC FLIGHT

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SUMMARY

An analysis was conducted to estimate the liquid-fuel temperatures and the amount of fuel evaporation that would be expected in operation of long-range supersonic aircraft at high altitudes. Two types of fuel losses were considered, the loss due to adiabatic evaporation during climb and the loss due to aerodynamic-heating effects. A flight plan was assumed, and heat-balance relations were employed to estimate the amount of heat transferred to fuel contained in a cylindrical fuselage. The aircraft operating range considered was: altitude, 60,000 to 90,000 feet; flight Mach number, 2 to 4; design flight time, $1\frac{1}{2}$ to 3 hours. The influence of flight speed, flight altitude, fuel withdrawal rate, tank size, tank pressurization, insulation, and initial fuel temperature on the evaporation of MIL-F-5624A grade JP-4 fuel was predicted. The influence of fuel volatility was determined by comparing the evaporation obtained with JP-4 fuel and that obtained with a lower volatility fuel, MIL-F-7914 grade JP-5.

The results of the analysis predicted, for an uninsulated tank, JP-4 fuel-boiling losses of 38 percent of the total fuel weight carried by the aircraft at the maximum flight speed and altitude considered. Fuel losses increased with increasing flight speed and flight time but decreased with increasing altitude. The fuel loss caused by adiabatic evaporation could be eliminated by ground refrigeration, tank pressurization, or the use of low volatility fuel. Fuel loss due to aerodynamic heating could be eliminated by a combination of insulation of a practical thickness and use of a low volatility fuel; however, if tank pressurization were employed without insulation, excessive pressures would be required. Without insulation, a maximum liquid-fuel temperature of 740° F would result if no fuel cooling occurred as a result of fuel evaporation; high liquid-fuel temperatures may create additional fuel handling problems.

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INTRODUCTION

The loss of aircraft fuel by evaporation from vented tanks can result in an appreciable reduction in flight range. Evaporation losses in aircraft fuel tanks operated below fuel-boiling altitudes have been studied and the results are presented in reference 1. Altitude-boiling losses, herein referred to as "adiabatic losses", have been empirically related to fuel volatility characteristics, fuel temperature, and operating altitude in reference 2. Some alleviation of the evaporation-loss problem has resulted from a reduction in jet-fuel volatility by changing from JP-3 to JP-4 specifications; however, a minimum volatility is still required to retain satisfactory engine starting characteristics and to assure adequate fuel availability.

Recent trends in aircraft development are toward operation at higher altitudes and increased flight speeds. Data of references 1 and 2 indicate that increased losses are to be expected at the higher altitudes; however, little data have been published to indicate the magnitude of losses sustained as a result of aerodynamic heating at high flight speeds. The purpose of this report is to present an estimation of expected liquid-fuel temperatures and of fuel-boiling losses at high altitudes and high flight speeds as determined by the analytical methods considered herein. The influence of flight speed, flight altitude, fuel withdrawal rate, tank size, tank pressurization, insulation, initial fuel temperature, and fuel volatility on the losses in an aircraft fuel tank moving at supersonic velocities is predicted. The analytical methods used herein were based on a heat-balance relation that includes ambient altitude pressures, aerodynamic heat-transfer rates, and physical characteristics of the fuel. Losses resulting from fuel slugging and foaming were not considered, because such losses are primarily a function of tank and vent design.

For the purpose of the analysis, the fuel tanks considered were assumed to occupy the total cross-sectional area of a right circular fuselage with heat being transferred from the boundary-layer air along the exterior lateral surface of the fuselage. Results obtained from this type of configuration would also describe losses occurring from externally mounted tanks. The tank capacities and flight times were chosen to be representative of the range of current and near future turbojet- and ram-jet-powered aircraft requirements. Flight Mach numbers of 2, 3, and 4; flight altitudes of 60,000 and 90,000 feet; initial fuel temperatures of 60° and 100° F; two fuels, MIL-F-5624A grade JP-4 and MIL-F-7914 grade JP-5; tank diameters of 3 and 6 feet; and two fuel withdrawal rates were studied. In addition, calculations were performed to indicate the reduction in fuel evaporation possible through the use of fuel tank pressurization and insulation. Comparison of the results of the analysis indicated the effects of the aforementioned variables on fuel losses and fuel temperature. Details of the methods of analysis used are presented.

This investigation was conducted at the NACA Lewis laboratory.

SYMBOLS

The following symbols are used in this report:

A	area for heat transfer, sq ft
C	constant
c_p	specific heat at constant pressure, Btu/(lb)($^\circ$F)
D	diameter, ft
Gr	Grashof number, $\frac{\beta \Delta T \rho^2 D^3 g}{\mu^2}$, dimensionless
g	acceleration due to gravity, 32.2 ft/sec^2
H	latent heat of vaporization, Btu/lb
h	heat-transfer coefficient for single resistance, Btu/(hr)(ft)^2($^\circ$F)
k	thermal conductivity, Btu/(hr)(ft)^2($^\circ$F/ft)
L	length, ft
Pr	Prandtl number, $c_p\mu/k$, dimensionless
Re	Reynolds number, $L\rho V/\mu$, dimensionless
T	temperature, $^\circ$F
U	over-all heat-transfer coefficient, Btu/(hr)(ft)^2($^\circ$F)
V	velocity, ft/sec
W	weight of fuel, lb
β	volumetric coefficient of thermal expansion
θ	time, hr
μ	coefficient of viscosity, lb/(ft)(sec) or lb/(ft)(hr)

ρ density, lb/ft^3

σ temperature recovery factor

Subscripts:

a air

ab boundary-layer air

am ambient air

av average

f fuel

0 total

v vapor

x distance from fuselage apex, ft

Δ increment

2 final

ANALYSIS

The effects of flight, design, and fuel variables on fuel temperature and evaporation losses were calculated from known heat-transfer and fuel-distillation relations. The assumptions, relations, and methods of calculation used to determine these effects are discussed.

Assumptions

Flight plan. - The flight plan was considered to have two phases: (1) a climb from take-off to the altitude considered, where the fuel depletion was due to evaporation at the reduced pressure; and (2) a constant velocity and constant altitude flight until the fuel was completely depleted by boiling due to aerodynamic heating and by withdrawal to the engine. Fuel consumption due to the factors mentioned in phase (2) was considered not to have occurred in phase (1), because of the short length of time required to attain the flight altitude, and because an auxiliary source of power was assumed for the thrust necessary for take-off and climb.

Fuel tank. - The fuselage configurations assumed for a basis of calculations are diagrammatically shown in figure 1. The fuel is contained in the fuselage, which occupies the total cross-sectional space rearward of the conical portion of the fuselage. The length-diameter ratio was 6 for the two tanks considered; the tanks had diameters of 6 and 3 feet. The insulation blanket was considered to cover the exterior of the fuselage and completely surround the lateral surface of the tank.

The following general assumptions pertaining to the exterior, the walls, and the interior conditions of the tank were made:

(a) The average heat-transfer coefficient from the boundary-layer air to the metal wall is described by an equation used to determine the rate of heat transfer across a flat plate in turbulent longitudinal flow; the ambient conditions at the boundary layer are given in the NACA standard atmosphere tables.

(b) There was no resistance to heat transfer through the metal tank walls and no transfer of heat to the fuel from any other direction or area except the wall area wetted by the liquid in the tank, excluding the area wetted at each end of the tank.

(c) The heat-transfer coefficient from the metal wall to the fuel is expressed by an equation governing turbulent free-convection flow on a vertical plate; liquid-vapor relations existing in the tank during aerodynamic heating were considered similar to an A.S.T.M. distillation.

Heat-Transfer Relations

The complete heat-balance equation representing the temperature change of the fuel has the form:

> Heat transferred from air stream to fuel tank + Heat transferred by solar and nocturnal irradiation to fuel tank = Sensible heat increase in liquid fuel + Sensible heat of vapor loss due to fuel boiling + Heat lost by radiation to surroundings

For the results presented herein the effects of irradiation to, and radiation from the fuel tanks were neglected. Calculations based on the methods presented in reference 3 indicated that the heat transferred to the fuel tank by irradiation would not significantly affect fuel temperatures. Also, with fuel boiling permitted, the metal temperatures of the tank walls would be sufficiently low to make the heat radiation to the surroundings an unimportant factor. In the case of pressurized fuel tanks, an error of 8 percent in the heat transferred to the fuel could result from neglecting the radiation effects at the maximum wall temperatures encountered.

The relations used in the analysis, as they are encountered in the flow of heat from the surrounding stratosphere into the fuel, are discussed in the following paragraphs.

Boundary-layer-air temperature. - The boundary-layer-air static temperature T_{ab} at the various Mach numbers considered was calculated from the following relation (ref. 4):

$$T_{ab} = T_{am} + \frac{\sigma V^2}{12,000} \tag{1}$$

The temperature recovery factor σ was assumed to be 0.885 for all Mach numbers and altitudes considered. Reference 5 indicates this to be an average value for bodies of revolution in the Mach number range considered; the limits of 0.885 ± 0.011 quoted in reference 5 would vary the value of the boundary-layer static temperature only ±1 percent. Also, the possible 60° variation in atmospheric ambient temperature, as reported in reference 6, would have a greater effect on boundary-layer temperature than would the possible variations in recovery factor. The boundary-layer-air temperature based on equation (1) is shown as a function of flight speed in figure 2.

Heat transfer through air film. - For all the analyses considered herein the average coefficient of heat transfer through the air film h_a was calculated from the following equation, which describes the transfer of heat across a flat plate in longitudinal turbulent flow (ref. 4):

$$\frac{h_a L_x}{k} = 0.037\left(\frac{L_x V \rho}{\mu}\right)^{0.8}\left(\frac{c_p \mu}{k}\right)^{1/3} \tag{2}$$

All air properties were evaluated at the average temperature between the fuselage-metal temperature and the boundary-layer-air static temperature. Reference 3 was used to obtain values of these air properties. The value of plate length L_x was 28 feet for the 6-foot-diameter tank and 19 feet for the 3-foot-diameter tank (fig. 1).

Values of the heat-transfer coefficient calculated from equation (2) were found to agree well with those calculated from an equation describing heat transfer across a body of revolution in turbulent flow (ref. 7). For example, the value of h_a for a 6-foot-diameter tank at an altitude of 90,000 feet and a flight Mach number of 3 is 5.3 Btu/(hr)(ft)^2($^{\circ}$F) for the body-of-revolution equation and 5.1 Btu/(hr)(ft)^2($^{\circ}$F) for the flat-plate equation (eq. (2)). This near equality of transfer coefficient would not necessarily exist for all tank sizes and conditions, but it does indicate reliability of the method used herein.

At the values of Reynolds number based on plate lengths L_x (approximately 6×10^6) existing along the fuselage walls at the flight speeds and altitudes assumed, the flow would probably be turbulent; however, it is possible that laminar-flow conditions could exist because of the smooth exterior surfaces and the cooling of these surfaces by the fuel. If laminar flow did exist, the value of heat-transfer coefficient would be obtained by the following equation (ref. 4):

$$\frac{h_a L_x}{k} = 0.331(\mathrm{Pr})^{1/3}(\mathrm{Re}_x)^{1/2} \tag{3}$$

Use of equation (3) would result in lowering the over-all heat-transfer rate approximately one order of magnitude, consequently the fuel would be heated at a rate approximately one-tenth of that calculated in the present analysis.

Heat transfer through liquid film. - The coefficient of heat transfer across the liquid-fuel film on the wetted metal surface h_f was calculated from the following equation, which describes turbulent free-convection heat flow across a vertical wall or a large cylinder (ref. 4):

$$\frac{h_f D}{k} = C(\mathrm{Pr}\cdot\mathrm{Gr})^{1/3} \tag{4}$$

The value of Grashof number was found to be greater than the critical value (10^9) for plate heights (tank diameters) greater than 1/2 foot; therefore, the boundary layer was considered turbulent at all times. No data were available to evaluate the constant C for liquid hydrocarbons; however, a value of 0.17 for water was reported in reference 4. Since the hydrocarbon fuels used in this analysis had viscosity and Prandtl number values similar to water, a value of $C = 0.2$ was chosen for all calculations in this report. A more rigorous evaluation of the fuel-film coefficient would be difficult because of the forced convection effect caused by fuel sloshing and withdrawal to the engine; also, the relative magnitudes of the fuel- and air-film coefficients are such that small errors in the fuel-coefficient value would not appreciably affect the value of over-all heat-transfer coefficient.

Equation (4) can be simplified to the following expression:

$$h_f = 11.86\left[\frac{(\rho_f)^2(c_{p,f})(k_f)^2\ \Delta T}{\mu_f}\right]^{1/3} \tag{5}$$

Equation (5) was used to calculate the liquid-film coefficient in all cases considered herein. The fuel properties were evaluated at the

average of the metal-wall and the bulk-fuel temperatures. Viscosity, specific gravity, and thermal conductivity were obtained from references 8, 9, and 10, respectively. A value of specific heat of 0.5 Btu per pound was assumed for both fuels. The ΔT term in equation (5) is the difference between the bulk-fuel temperature and the metal-wall temperature. Calculations indicated that, as the fuel temperature increased during flight, the resulting variation in the value of ΔT would not greatly affect the film heat-transfer coefficient.

Over-all heat-transfer coefficient. - The over-all heat-transfer coefficient U was determined from the relation:

$$U = \frac{1}{\dfrac{1}{h_a} + \dfrac{1}{h_f}} \tag{6}$$

The values of h_a and h_f in equation (6) were determined at a value of metal-wall temperature that satisfied the equality of heat flow from air to metal wall and from metal wall to fuel. That is, the amount of heat transferred, on an equal area basis, to the metal wall was equal to the amount of heat transferred away from the metal wall.

Heat-transfer area. - A plot of the relation between the ratio of wetted area to total area and the ratio of partially filled volume to total volume for a right circular cylinder, as determined by the equation

$$\frac{W}{W_0} = \frac{A}{A_0} - \frac{1}{2\pi} \sin\left(\frac{2\pi A}{A_0}\right) \tag{7}$$

is presented in figure 3. This figure was used to obtain the wetted area, or heat-transfer area, for the amount of fuel contained in the tank at any flight time.

Fuel Distillation Relations

The fuel properties pertinent to the analysis are shown in table I. The fuel evaporated during the climb to the altitude considered (phase (1) of the flight plan) was calculated as adiabatic fuel loss by the methods outlined in reference 2. Sea-level-pressure A.S.T.M. distillation curves were estimated for the fuel stocks remaining after the adiabatic fuel loss occurred in phase (1). The A.S.T.M. curves used in calculations concerning phase (2) of the flight plan were obtained by extrapolating the sea-level-pressure A.S.T.M. curves to the ambient pressure at the altitude considered (ref. 9, p. 127). A.S.T.M. curves for the different conditions of analysis are presented in figures 4 and 5 for the JP-4

and JP-5 fuels, respectively. The fact that the residual fuel temperature after adiabatic loss was within 2° F of the initial boiling temperature of the extrapolated A.S.T.M. distillation curve indicates that these two different methods of finding residual fuel temperature provided consistent results.

Method of Calculation

The complete heat-balance equation, in which radiation effects are neglected, is:

Heat transferred from air stream to fuel = Heat absorbed by liquid fuel + Sensible heat of vapor loss due to fuel boiling

or

$$UA_{av}(T_{ab} - T_{f,av})\Delta\theta = W_{av}\, c_p(T_{f,2} - T_{f,av}) + H\Delta W_v \tag{8}$$

Fuel temperature and the fuel evaporation were determined by use of equation (8) for small intervals of flight time. A time interval $\Delta\theta$ equal to 1/12 of the total flight time, assuming no fuel loss, was selected to provide sufficient precision for all calculations. Average values existing in the time interval were substituted into equation (8), and evaluation of the final fuel temperature and the percentage of fuel evaporated for the time interval was obtained when equation (8) was balanced. The terms of equation (8) are discussed in the following paragraphs.

<u>Heat transferred from air stream to fuel $UA_{av}(T_{ab} - T_{f,av})\Delta\theta$.</u> - The over-all heat-transfer coefficient U is determined by use of equation (6); the average heat-transfer area A_{av}, from figure 3. The temperature driving potential $T_{ab} - T_{f,av}$ is the difference between the boundary-layer-air temperature, as determined by equation (1), and the average liquid-fuel temperature during the chosen time interval $\Delta\theta$.

<u>Sensible heat increase of liquid fuel $W_{av}\, c_p(T_{f,2} - T_{f,av})$.</u> - The average weight of fuel W_{av} is the arithmetic mean of the initial and final mass of fuel in the tank for the time interval. The specific heat c_p was assumed to be constant for both fuels and all temperatures. The temperature increase of the liquid fuel $T_{f,2} - T_{f,av}$ is the difference between the final fuel temperature, an unknown, and the average fuel temperature.

<u>Sensible heat of vapor loss $H\Delta W_v$.</u> - The latent heat of vaporization H was estimated by use of the graph on page 96 of reference 10 for each

pressure-altitude condition and each fuel considered. The weight of fuel evaporated ΔW_v during the time interval is an unknown.

Solution of the complete heat-balance equation to determine the amount of fuel evaporated ΔW_v and the liquid-fuel temperature at the end of the time interval $T_{f,2}$ was obtained by a method of successive approximations. The value of ΔW_v during the time interval was assumed, and the corresponding $T_{f,2}$ was read from the appropriate A.S.T.M. curve. The initial fuel temperature used to determine the average fuel temperature for the first time interval was equal to the residual fuel temperature after adiabatic loss in phase (1) of the flight plan. By substituting these values and the other required numerical values into equation (8), the accuracy of the first approximation of ΔW_v was determined. A second assumption was then made for ΔW_v, and the procedure was repeated until values of ΔW_v and $T_{f,2}$ were obtained which simultaneously satisfied the heat-transfer relation represented by equation (8) and the appropriate A.S.T.M. curve of figure 4 or 5. The calculation was repeated for each succeeding time interval until the fuel evaporated plus the amount withdrawn to the engine exhausted the original fuel supply. To obtain more precise results, smaller time intervals than 1/12 of the flight time were chosen when the amount of liquid fuel left in the tank was approximately 10 percent of the initial amount.

Calculations were made for both types of fuel heating, with and without evaporation. When the tank was pressurized to eliminate fuel evaporation, the sensible heat of vapor loss term in equation (8) was equal to zero and the final liquid fuel temperature $T_{f,2}$ was determined by successive approximation.

RESULTS AND DISCUSSION

A summary of calculation variables and results obtained from the analysis of aerodynamic fuel heating is presented in table II. The discussion of the effects of the various primary variables on aerodynamic fuel heating is divided into three major sections - flight variables, design variables, and fuel variables.

Flight Variables

The effects of flight speed, flight altitude, fuel withdrawal rate, and ambient-air temperature on JP-4 fuel temperature and fuel losses are presented for a 6-foot uninsulated integral tank 36 feet in length and containing 50,000 pounds of fuel. The initial fuel temperature was 60° F at sea level.

Flight speed. - Fuel evaporation and residual fuel temperature are plotted in figure 6 as functions of flight time for three flight Mach numbers - 2, 3, and 4. The flight altitude represented is 90,000 feet and the fuel withdrawal rate to the engine is $33\frac{1}{3}$ percent of the initial fuel weight per hour. The residual fuel temperature at 90,000 feet after cooling resulting from adiabatic fuel loss is 30° F. As shown in figure 2, the boundary-layer air does not attain a temperature of 30° F until a Mach number of approximately 1.2 is reached; no fuel boiling will occur as a result of aerodynamic heating at lower flight speeds regardless of heat-transfer rates. It is noted that evaporation losses increased with flight time and with flight speed. The total fuel-evaporation loss is a maximum of 37.5 percent at a Mach number of 4; this loss represents the percentage reduction in expected flight time due to fuel boiling. Of the total loss encountered at a Mach number of 4, approximately four-fifths was attributable to aerodynamic heating alone; the accompanying residual fuel temperatures are presented in figure 6(b). The maximum temperature predicted, 117° F, would not be expected to introduce any additional fuel handling problems. The observed increases in evaporation loss and fuel temperature with flight speed resulted from an increase in the boundary-layer temperature and in the over-all heat-transfer coefficient. This effect would be present regardless of altitude, tank size, type of fuel, or any other variable, for a given flight period.

Flight altitude. - Fuel evaporation and residual fuel temperature are plotted in figure 7 as functions of flight time for pressure-altitudes of 60,000 and 90,000 feet, a flight Mach number of 3, and a fuel withdrawal rate of $33\frac{1}{3}$ percent of the initial fuel weight per hour. At 60,000 feet, the adiabatic fuel loss is 1 percent and at 90,000 feet, 8 percent; however, the loss due to aerodynamic heating alone is 35.5 percent at 60,000 feet as compared with 17.5 percent at 90,000 feet. As shown in figure 7(b), the final residual fuel temperature encountered at 60,000 feet altitude is 170° F, or approximately 100° F higher than that encountered at 90,000 feet altitude. The increase in total fuel loss and in residual-liquid-fuel temperature at the lower altitude is caused by the increase in heat-transfer coefficient resulting from the higher ambient-air density at 60,000 feet. The air-film coefficient (eq. (2)), which is the controlling resistance, varies directly with density to the 0.8 power. Since the increase in ambient-air density is approximately fourfold between 90,000 and 60,000 feet, the corresponding over-all heat-transfer rate is increased approximately threefold. These results indicate that for long-range aircraft operating in the stratosphere, if the flight time is sufficient to cause appreciable fuel loss due to aerodynamic heating, minimum total fuel loss would be experienced at the maximum flight altitude.

Fuel withdrawal rate. - Fuel evaporation and residual fuel temperature are plotted in figure 8 as functions of flight time for two fuel withdrawal rates - $33\frac{1}{3}$ and $66\frac{2}{3}$ percent of the initial fuel weight per hour. The flight Mach number is 3 and the flight altitude, 90,000 feet. The percentage fuel evaporated by aerodynamic heating at the lower withdrawal rate is 17.5 percent and at the higher rate, 9.5 percent. For a given tank size, the effect of varying withdrawal rate is to vary the flight time and, consequently, the total amount of heat transferred to the fuel. However, doubling the withdrawal rate, or halving the flight time, did not proportionately reduce the percentage of fuel lost due to aerodynamic heating. Fuel loss would be directly proportional to flight time only if the A.S.T.M. distillation curves (fig. 4(a)) and the volume-area relation curve of the tank (fig. 3) were linear. As an approximation, the percentage of fuel evaporated may be considered proportional to exposure time.

Ambient-air temperature. - The ambient-air temperature used for the altitudes considered in the results presented was -67° F, the value given by NACA standard-atmosphere tables. The observed variation in this temperature of approximately $\pm 60^\circ$ F (ref. 6) would alter the boundary-layer temperature and the air-film heat-transfer coefficient. At an altitude of 90,000 feet and a Mach number of 3, variations in the amount of fuel lost due to aerodynamic heating of approximately ± 2 to ± 3 percent of the total fuel weight would result.

Design Variables

The effects of tank size, tank pressurization, and insulation on JP-4 fuel temperature and evaporation losses are presented for a flight altitude of 90,000 feet. The effects of tank size and insulation were determined for a flight Mach number of 3, and the effects of tank pressurization, for flight Mach numbers of 2, 3, and 4. The tank pressures required to eliminate fuel boiling at the three Mach numbers for both JP-4 and JP-5 fuels are included. The sea-level initial fuel temperature was 60° F and the fuel withdrawal rate, $33\frac{1}{3}$ percent of initial fuel weight per hour.

Tank size. - Aerodynamic-heating effects in two tanks, each having a length to diameter ratio of 6, were determined. The larger tank had a diameter of 6 feet and a capacity of 50,000 pounds of fuel; the smaller tank had a diameter of 3 feet and a capacity of 6250 pounds of fuel. The fuel evaporation and residual fuel temperatures are plotted in figure 9 as functions of flight time for the two tanks. The adiabatic fuel loss, of course, is identical percentage-wise for both tanks. The

smaller tank sustained 30.3-percent fuel loss by aerodynamic heating; the larger tank, 17.5 percent. The over-all heat-transfer coefficient was increased slightly, from 4.65 to 5.02 Btu/(hr)(ft)^2($^{\circ}$F), by decreasing the tank diameter from 6 to 3 feet (table II). However, the increase in fuel evaporation was primarily due to the increase in lateral surface area (heat-transfer area) per pound of fuel. For the 6-foot-diameter tank, the ratio of total lateral area per pound of fuel was 0.0135; for the 3-foot-diameter tank, 0.027. These results indicate that the most important factor in aerodynamic heat transfer into the fuel, as influenced by tank configuration, is the value of transfer area per pound of fuel.

Tank pressurization. - Fuel-evaporation-loss curves for tank pressures of 0, 5, 50, and 100 pounds per square inch above ambient for the 6-foot-diameter tank and three flight speeds are presented in figure 10. The same data are plotted in figure 11 where percentage of fuel evaporated is shown as a function of flight speed. At a tank pressure of 5 pounds per square inch, the fuel lost by adiabatic evaporation at an altitude of 90,000 feet was completely eliminated, and the loss due to aerodynamic heating at a Mach number of 2 was negligible. At the higher flight speeds, a pressure of 5 pounds per square inch did not have a large effect on fuel loss due to aerodynamic heating; at a Mach number of 3 the reduction was 4.3 percent of the total fuel weight; and at a Mach number of 4, 2 percent of the total fuel weight. The total reduction in fuel-weight loss due to both aerodynamic heating and adiabatic loss during climb, however, was appreciable since losses due to the latter effect were eliminated.

The tank pressures required to eliminate fuel evaporation with both JP-4 and JP-5 fuels at Mach numbers from 2 to 4 are shown in figure 12. For the JP-4 fuel, a pressure of 325 pounds per square inch is required to eliminate fuel evaporation at a Mach number of 3, and 3400 pounds per square inch is required at a Mach number of 4. For the less volatile JP-5 fuel, the pressure required is 25 pounds at a Mach number of 3 and 500 pounds, at a Mach number of 4. The results presented in figures 11 and 12 indicate that a pressure differential of 5 pounds per square inch across the tank walls will minimize aerodynamic heating effects on JP-4 fuel at high altitudes in the flight Mach number range of 1 to 2; excessive tank pressures would be required at higher flight speeds. An appreciable reduction in the required tank pressurization results from a reduction in fuel volatility.

The liquid-fuel temperature attained in the 6-foot-diameter tank pressurized sufficiently to prevent evaporation is presented in figure 13 as a function of flight time. The liquid-fuel temperatures at the ends of the flights are 145°, 390°, and 740° F for Mach numbers of 2, 3, and 4, respectively. It is of interest to note that a change in

liquid-fuel temperature from 60° to 400° F may result in a decrease in fuel density of approximately 25° A.P.I. High fuel temperatures and attendant wide variations in fuel density during flight may cause malfunctioning of the engine fuel systems, especially if the metering controls operate on a fuel-volume basis. Also, the self-sealing feature of fuel tanks (considering present methods) would be inoperative with an effective amount of tank pressurization.

Insulation. - The effect of tank insulation on fuel evaporation due to aerodynamic heating is shown in figure 14. The insulation blanket considered had a thermal conductance of 0.5 Btu/(hr)$(ft)^2$(°F), which corresponds to a layer of cork approximately 1/2-inch thick with a thermal conductivity of 0.025 Btu/(hr)$(ft)^2$(°F/ft). This insulation added enough resistance to heat flow to lower the over-all heat-transfer coefficient approximately one order of magnitude at the 90,000 feet altitude - Mach number of 3 condition considered (table II). The amount of fuel evaporation due to aerodynamic heating was decreased from 17.5 to 2.2 percent by the addition of the insulation, as shown in figure 14.

Results are presented in reference 11 for a tank contained inside a fuselage with insulation having a thermal conductance value of 1 Btu/(hr)$(ft)^2$(°F). With the aircraft assumed to be traveling over a given flight path at a Mach number of 2.75, the maximum heat-transfer rate was reduced from 900 Btu per second to 80 Btu per second by the addition of insulation. The calculated liquid-fuel temperature at the end of the flight was reduced from 350° to 130° F. The results shown in reference 11 and in the present report indicate that use of insulation blankets of practical thickness may nearly eliminate fuel evaporation due to aerodynamic heating at flight speeds up to a Mach number of 3 at high altitude.

Fuel Variables

The effect of initial fuel temperature at sea level on fuel losses and residual fuel temperature and the effect of fuel volatility on fuel losses were determined for the 6-foot-diameter uninsulated tank. The flight Mach number considered was 3; the altitude, 90,000 feet; and the fuel withdrawal rate, $33\frac{1}{3}$ percent of the initial fuel weight per hour.

Initial fuel temperature. - The amount of JP-4 fuel evaporated and the residual fuel temperature for initial fuel temperatures of 60° and 100° F are shown in figure 15 as functions of flight time. The amount of adiabatic fuel loss at an altitude of 90,000 feet was 8 percent for the 60° F initial fuel temperature and 13.5 percent for the 100° F initial fuel temperature, a difference of 5.5 percent. The fuel loss due to aerodynamic heating was less by 1 percent of the total fuel weight for the higher initial fuel temperature; thus, the total loss

was 4.5 percent greater. Aerodynamic heating was reduced somewhat because of the smaller temperature differential between the boundary-layer air and the fuel. If ground refrigeration were used to precool the JP-4 fuel before flight, a fuel temperature of -3° F would be necessary to completely eliminate the 8 percent loss due to adiabatic evaporation at an altitude of 90,000 feet. However, not all of this 8 percent would be made available for engine consumption since it is estimated that the loss due to aerodynamic heating would increase approximately 2 to 3 percent of the initial fuel weight because of the increased temperature differential between the boundary-layer-air and bulk-fuel temperature.

Fuel volatility. - A comparison of fuel-evaporation losses with JP-4 and JP-5 fuels at an initial temperature of 60° F is presented in figure 16. The boiling ranges of both fuels are presented in table I. With the low volatility JP-5 fuel no adiabatic loss was encountered at an altitude of 90,000 feet. Also, 1 hour of flight time elapsed before the JP-5 liquid fuel reached its initial boiling temperature. A total loss of 10.5 percent occurred with the JP-5 fuel; entirely as a result of aerodynamic heating. With the JP-4 fuel, 8 percent of the total fuel was lost because of adiabatic evaporation and 17.5 percent because of aerodynamic-heating effects; thus, there was a total loss of 25.5 percent. These trends indicate that the use of low volatility fuels will substantially decrease fuel-boiling losses during long-range supersonic flight but will not eliminate the problem. In addition, the wide variations in fuel temperature and density caused by the use of lower volatility fuels would lead to fuel metering and control systems difficulties.

CONCLUDING REMARKS

An analysis was conducted to determine the fuel temperatures and the fuel-evaporation losses encountered in supersonic long-range flight at altitudes of 60,000 to 90,000 feet and at flight Mach numbers from 2 to 4. At the maximum flight speed and altitude considered, the analysis predicted evaporation losses of 38 percent of the total fuel weight with MIL-F-5624A grade JP-4 fuel. These fuel losses would be maximum at highest flight speeds, longest flight times, and lowest altitudes.

The results indicated that ground refrigeration could be employed to eliminate fuel loss due to adiabatic evaporation during climb, but not all of the fuel saved would be available to the engine, since the fuel loss due to aerodynamic heating was found to increase with decreasing initial fuel temperature. A combination of insulation, with a thermal conductance of 0.5 Btu/(hr)(ft)^2(°F), and low volatility fuel, such as MIL-F-7914 grade JP-5, could be used to eliminate fuel evaporation for the flight conditions studied. If higher fuel volatility were required

to meet the demands of engine performance and fuel availability, a tank pressurization of 5 pounds per square inch above ambient used in conjunction with tank insulation would have the same effect. If tank pressurization were employed without insulation, the loss due to adiabatic evaporation could be eliminated, but excessive pressures would be required to eliminate the loss due to aerodynamic heating.

Use of tank pressurization and low volatility fuels without insulation could result in maximum liquid-fuel temperatures of 740° F at a Mach number of 4 and an altitude of 90,000 feet, if no fuel cooling occurred as a result of fuel evaporation. These high fuel temperatures and the accompanying wide variations in fuel density during flight may affect engine fuel metering and control systems.

It should be realized that the trends presented herein were computed from an analysis containing a number of restrictive assumptions; the results should be considered qualitative. The flight assumptions made restrict the analysis to the case of a long-range supersonic aircraft; short-range interceptor-type aircraft may not encounter the rather severe aerodynamic-heating effects described herein. Also, in the analysis, only fuel losses due to evaporation were considered; losses due to foaming and slugging, which may be considerable, were neglected.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, May 18, 1953

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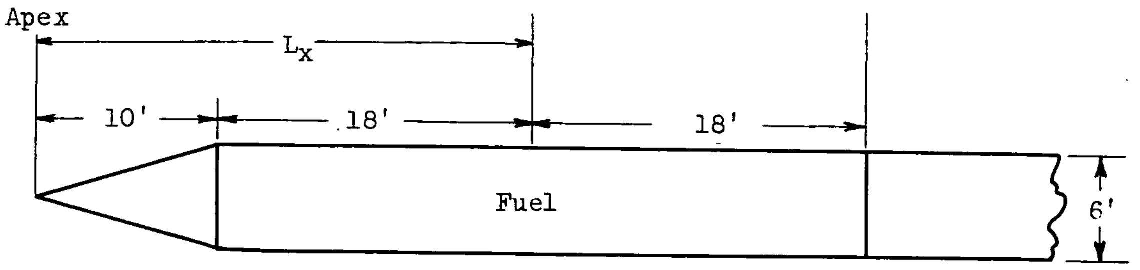
TABLE I. - PROPERTIES OF FUELS

Fuel properties	MIL-F-5624A grade JP-4	MIL-F-7914 grade JP-5
A.S.T.M. distillation D86-46, $^{\circ}$F		
Initial boiling point	135	357
Percentage evaporated		
5	176	371
10	210	375
20	245	385
30	279	393
40	311	402
50	345	411
60	378	421
70	412	433
80	445	448
90	483	464
Final boiling point	568	502
Gravity		
$^{\circ}$A.P.I.	47.8	43.7
Specific, $60^{\circ}/60^{\circ}$ F	0.789	0.808
Reid vapor pressure, lb/sq in.	2.4	0.2
Specific heat, Btu/lb	0.5	0.5
Viscosity (centistokes at 60° F), ref. 8	1.15	1.85
10-Percent-point slope, $^{\circ}$F/percent	5.1	1.1
Coefficient of thermal expansion, ref. 9	0.0005	0.0005

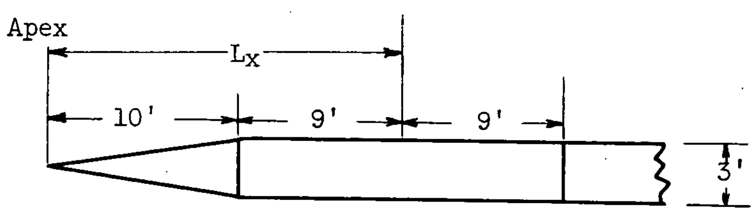
TABLE II. - CALCULATION VARIABLES AND RESULTS OBTAINED

Mach number	Flight altitude, ft	Design flight time with no vaporization, hr	Type of fuel	Total fuel weight, w_O, lb	Boundary air layer temperature, T_{ab}, ^{o}F	Over-all heat-transfer coefficient, U, Btu/(hr)(sq ft)(^{o}F)	Tank metal-wall temperature at altitude end of phase 1, ^{o}F	Tank diameter, D, ft	Tank length-diameter ratio	Total heat-transfer area, A_O, sq ft	Initial fuel temperature at sea level, ^{o}F	Latent heat of vaporization, H, Btu/lb (a)	Tank pressure above ambient, lb/sq in.	Fuel temperature at altitude condition, ^{o}F	Weight of fuel evaporated adiabatically, percent	Weight of fuel evaporated by aerodynamic heating, percent	Total weight of fuel evaporated, percent	Actual flight time, percent of design flight time
2	90x10³	3	JP-4	50x10³	213	3.59	45	6	6	679	60	162	0	30	8	5.3	13.3	86.7
3	90	3	JP-4	50	563	4.65	70	6	6	679	60	160	0	30	8	17.5	25.5	74.5
4	90	3	JP-4	50	1051	4.94	100	6	6	679	60	158	0	30	8	29.5	37.5	62.5
3	60	3	JP-4	50	563	12.9	130	6	6	679	60	158	0	50	1	35.5	36.5	63.5
3	90	1.5	JP-4	50	563	4.65	70	6	6	679	60	160	0	30	8	9.5	17.5	82.5
3	90	3	JP-4	6.25	563	5.02	75	3	6	169.5	60	160	0	30	8	30.3	38.3	61.7
2	90	3	JP-4	50	213	3.59	---	6	6	679	60	155	5.00	87	0	1.8	1.8	98.2
3	90	3	JP-4	50	563	4.65	---	6	6	679	60	147	5.00	87	0	13.2	13.2	86.8
4	90	3	JP-4	50	1051	4.94	---	6	6	679	60	144	5.00	87	0	27.5	27.5	72.5
3	90	3	JP-4	50	563	4.65	---	6	6	679	60	128	50	215	0	4.6	4.6	95.4
4	90	3	JP-4	50	1051	4.94	---	6	6	679	60	122	50	215	0	19.0	19.0	81.0
3	90	3	JP-4	50	563	4.65	---	6	6	679	60	114	100	275	0	2.0	2.0	98.0
4	90	3	JP-4	50	1051	4.94	---	6	6	679	60	108	100	275	0	16.0	16.0	84.0
3	90	3	JP-4	50	563	0.457[b]	40	6	6	679	60	162	0	30	8	2.2	10.2	89.8
3	90	3	JP-4	50	563	4.65	90	6	6	679	100	157	0	52	13.5	16.5	30.0	70.0
3	90	3	JP-5	51.1	563	4.57	100	6	6	679	60	142	0	154	0	10.5	10.5	89.5

[a]Ref. 10.
[b]Insulated fuel tank.



(a) Tank diameter, 6 feet.



(b) Tank diameter, 3 feet.

Figure 1. - Diagrammatic sketch of assumed fuselage configurations showing fuel-tank location.

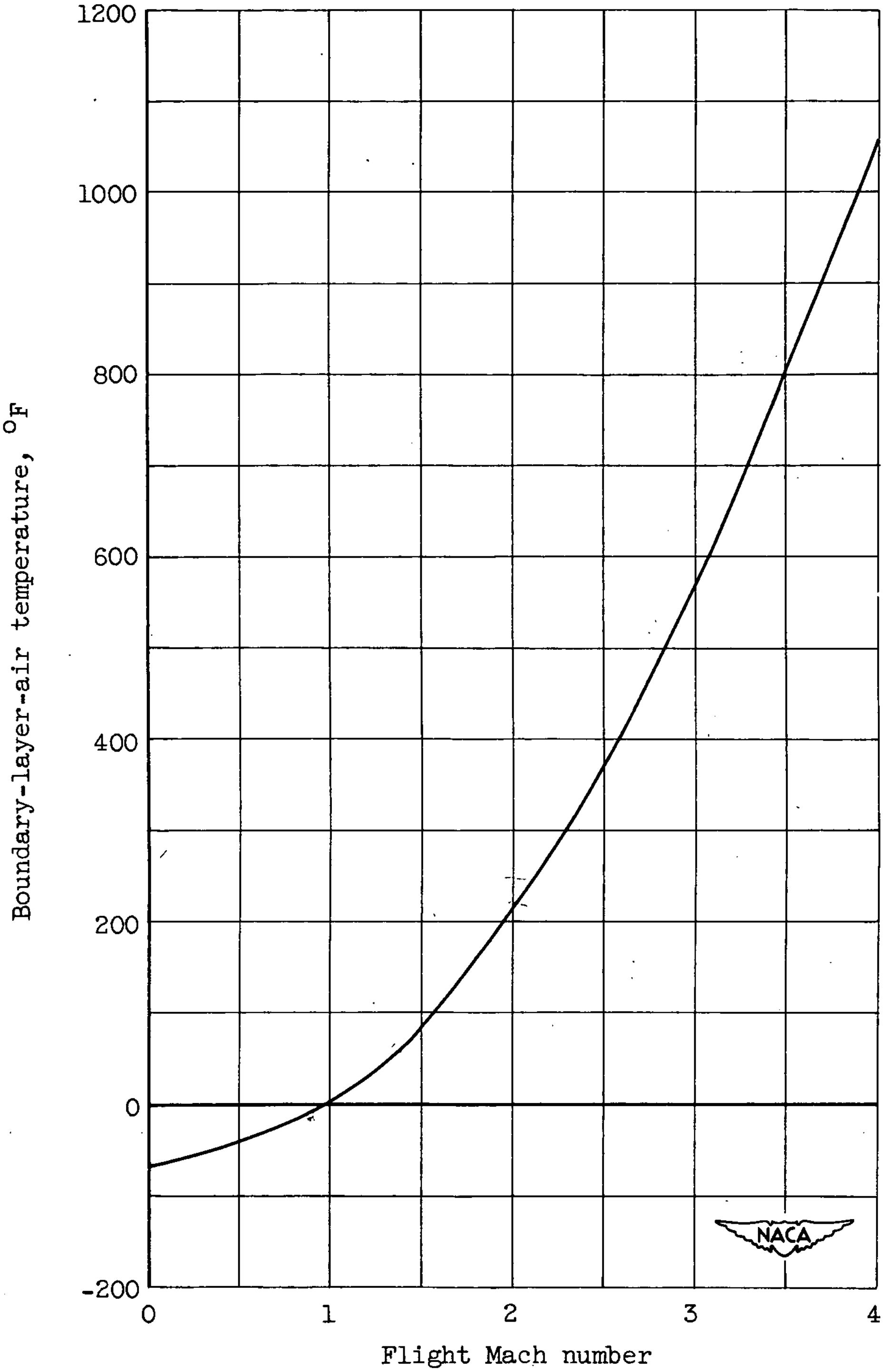


Figure 2. - Variation of boundary-layer-air temperature with Mach number. Recovery factor, 0.885; ambient-air temperature, -67° F.

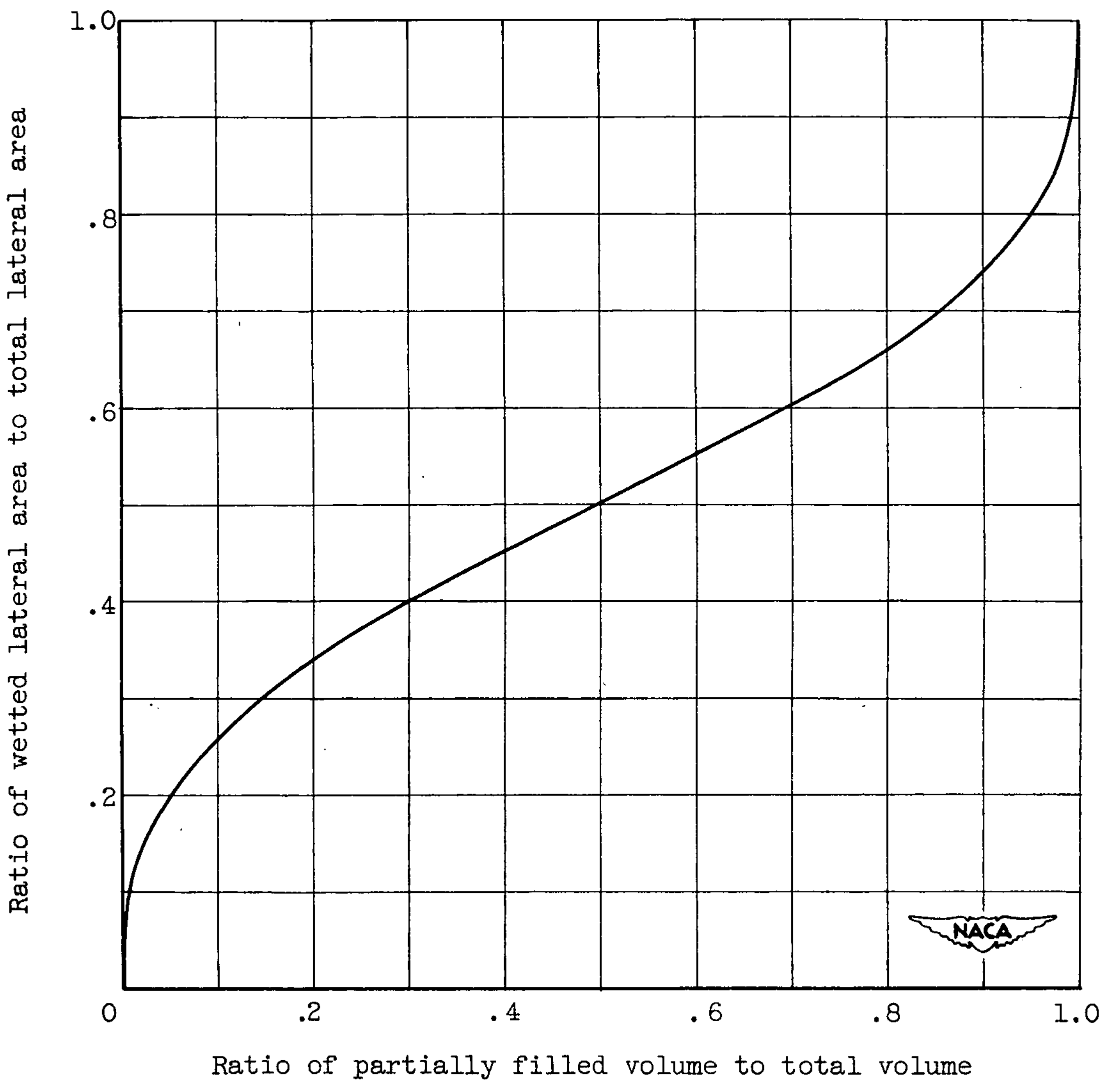
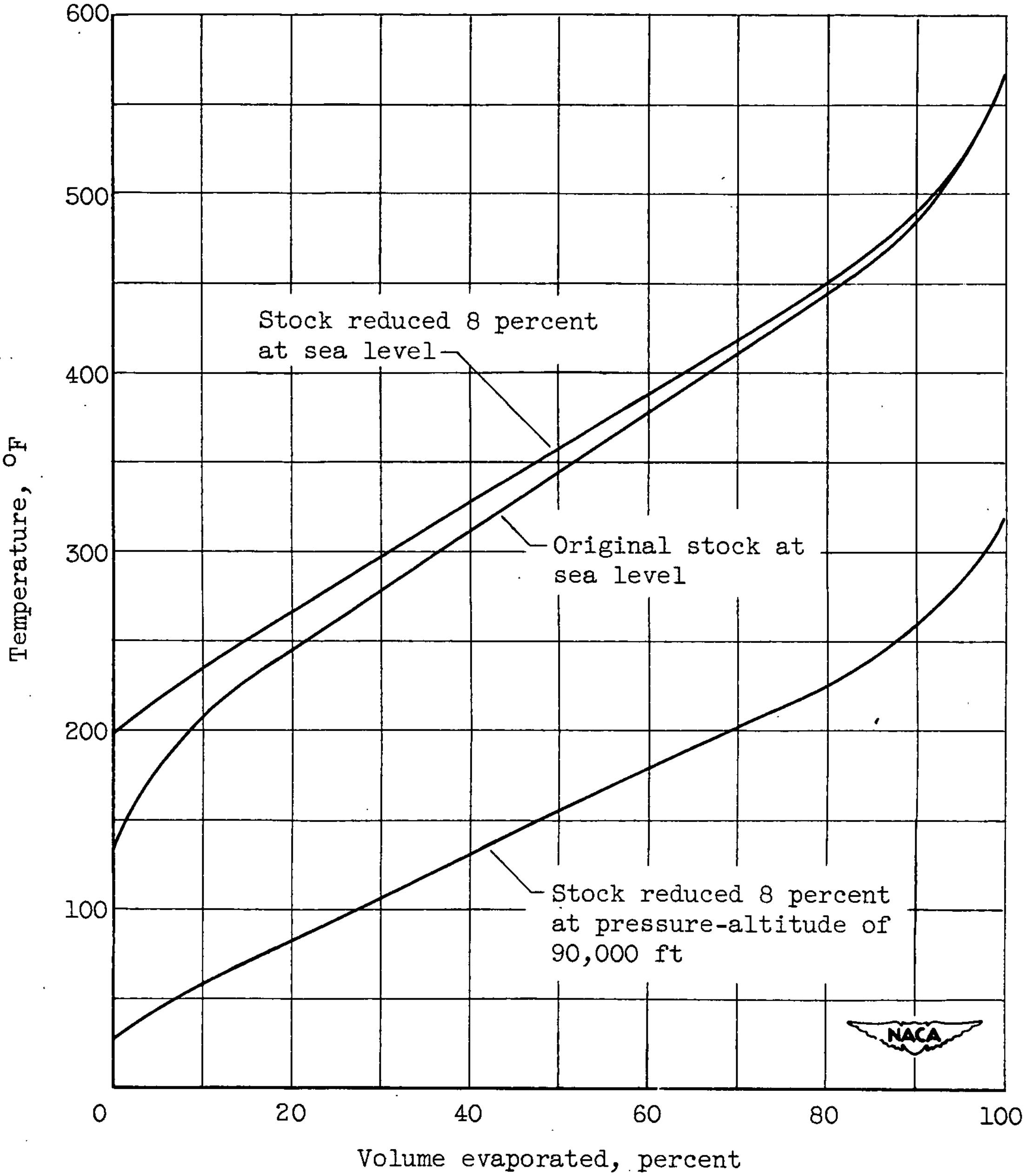
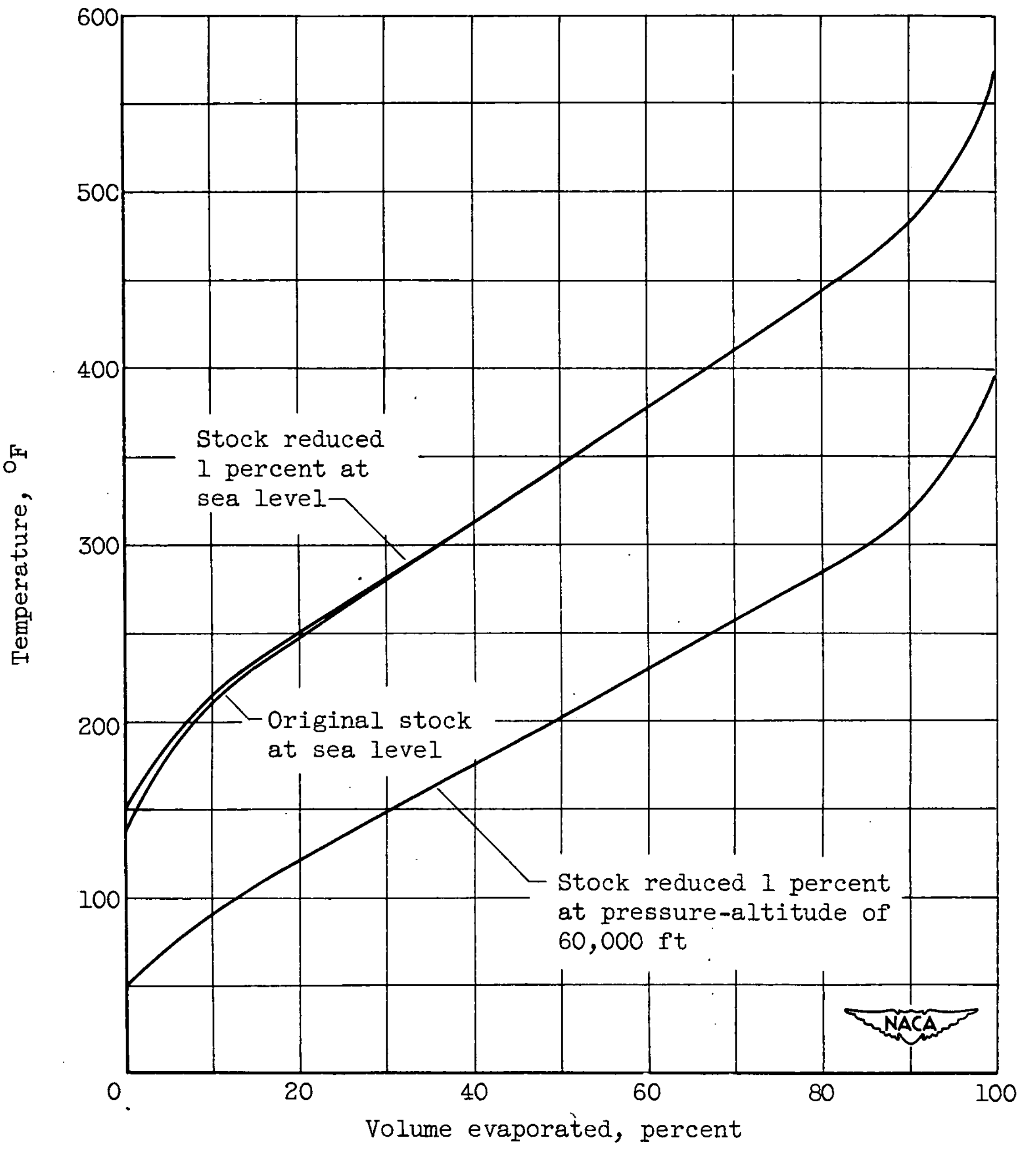


Figure 3. - Relation between ratio of wetted lateral area to total lateral area and ratio of partially filled volume to total volume of right circular cylinder with axis horizontal.



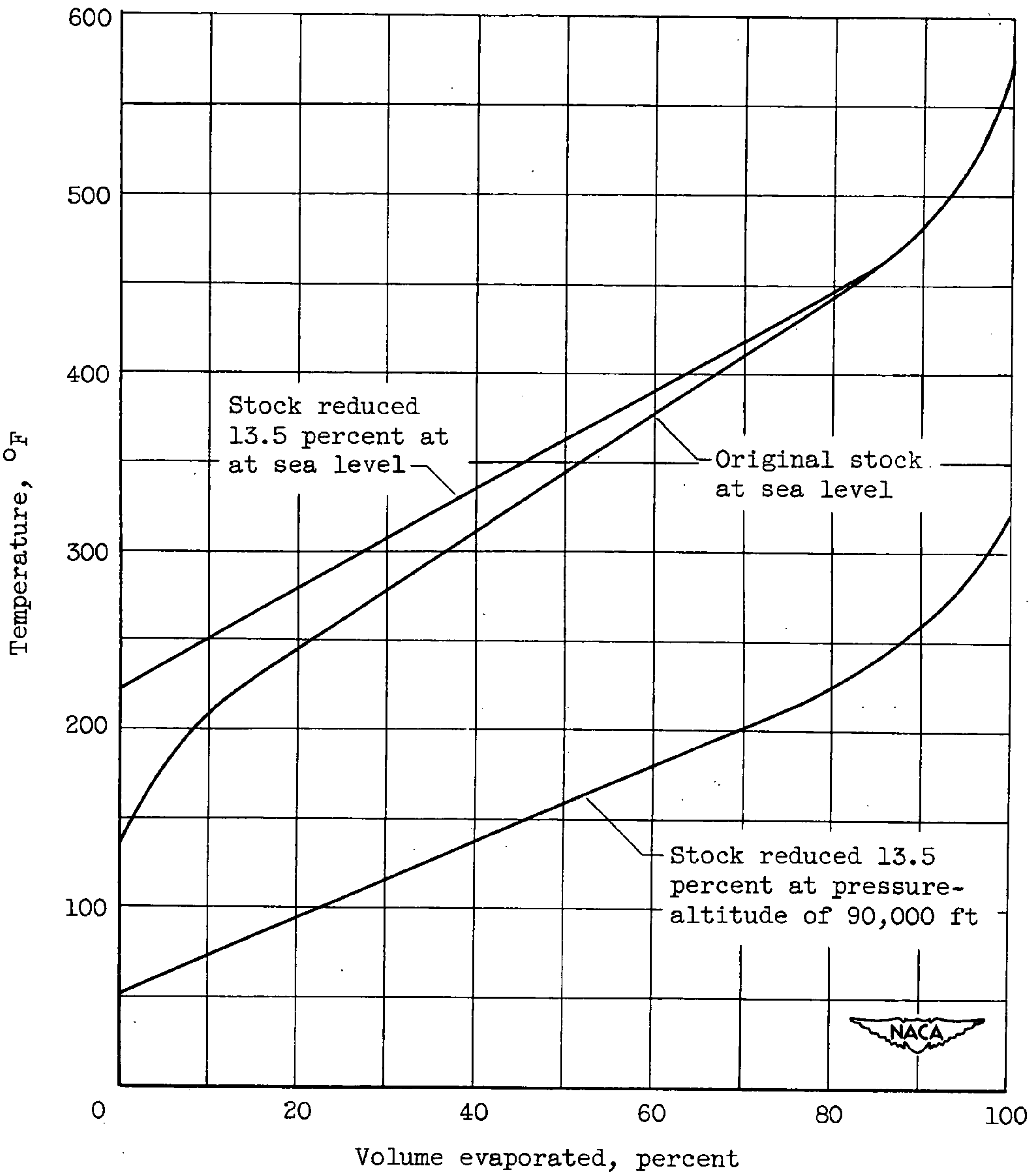
(a) Pressure-altitude, 90,000 feet; initial fuel temperature, 60° F; adiabatic loss, 8 percent.

Figure 4. - A.S.T.M. distillation curves for MIL-F-5624A grade JP-4 fuel at various analysis conditions.



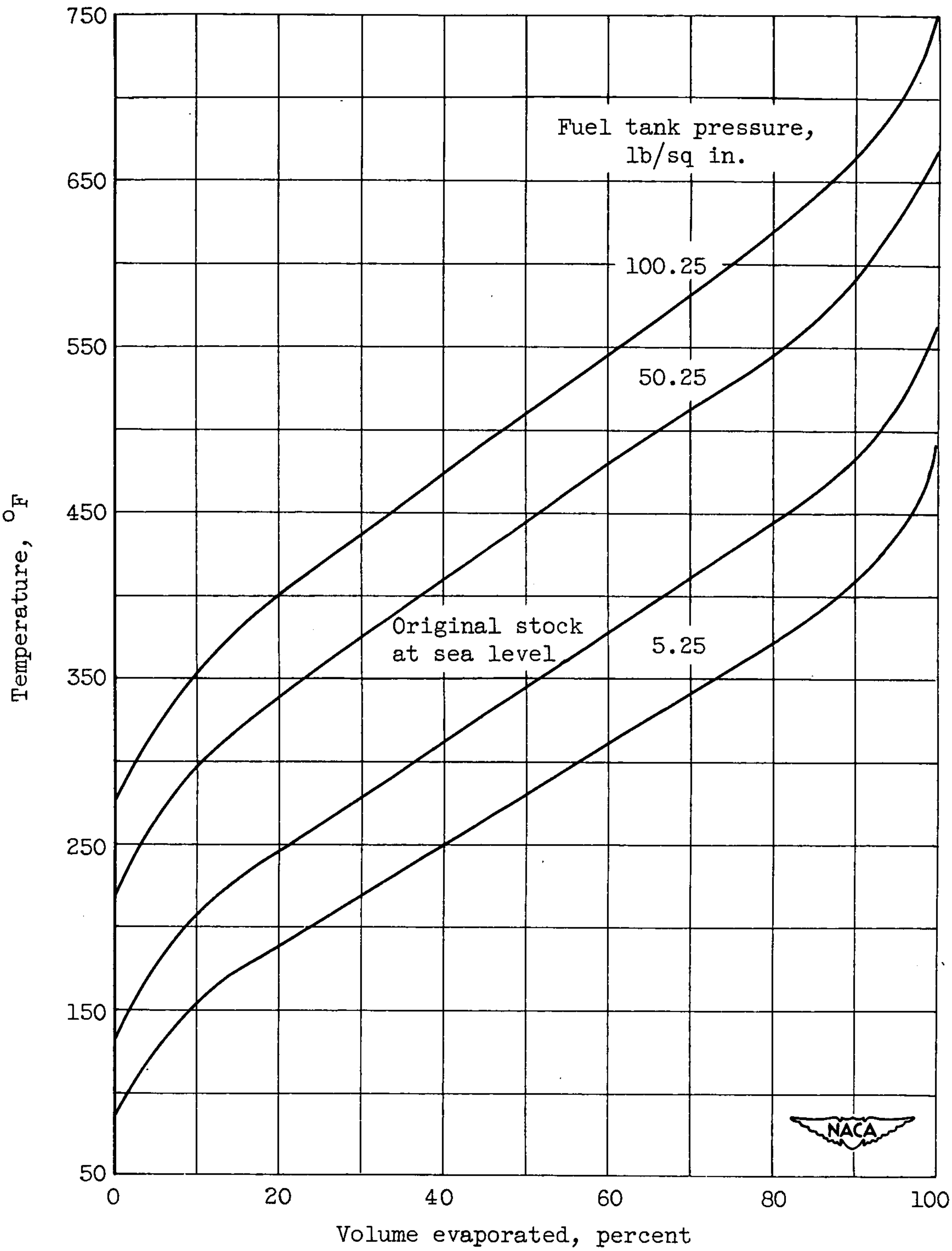
(b) Pressure-altitude, 60,000 feet; initial fuel temperature, 60° F; adiabatic loss, 1 percent.

Figure 4. - Continued. A.S.T.M. distillation curves for MIL-F-5624A grade JP-4 fuel at various analysis conditions.



(c) Pressure-altitude, 90,000 feet; initial fuel temperature, 100° F; adiabatic loss, 13.5 percent.

Figure 4. - Continued. A.S.T.M. distillation curves for MIL-F-5624A grade JP-4 fuel at various analysis conditions.



(d) Effect of tank pressurization. Pressure-altitude, 90,000 feet; initial fuel temperature, 60° F; no adiabatic loss.

Figure 4. - Concluded. A.S.T.M. distillation curves for MIL-F-5624A grade JP-4 fuel at various analysis conditions.

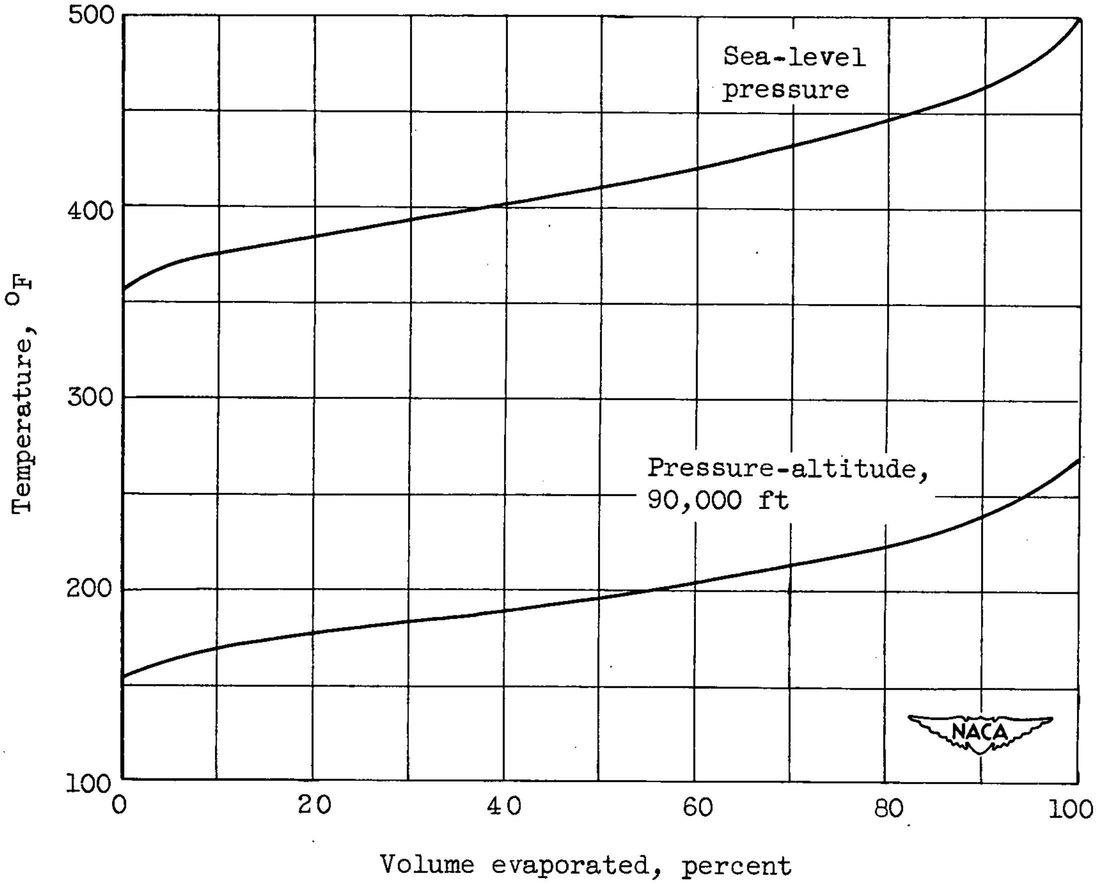
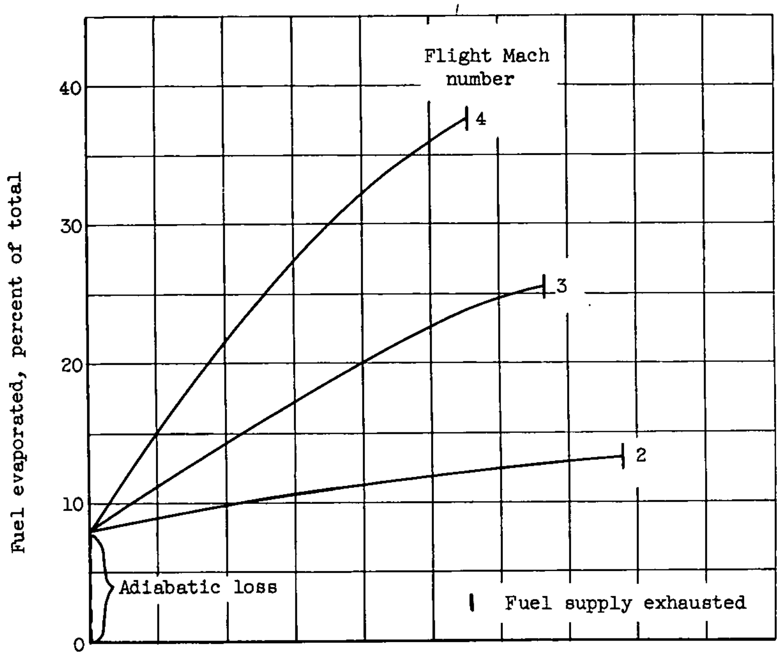
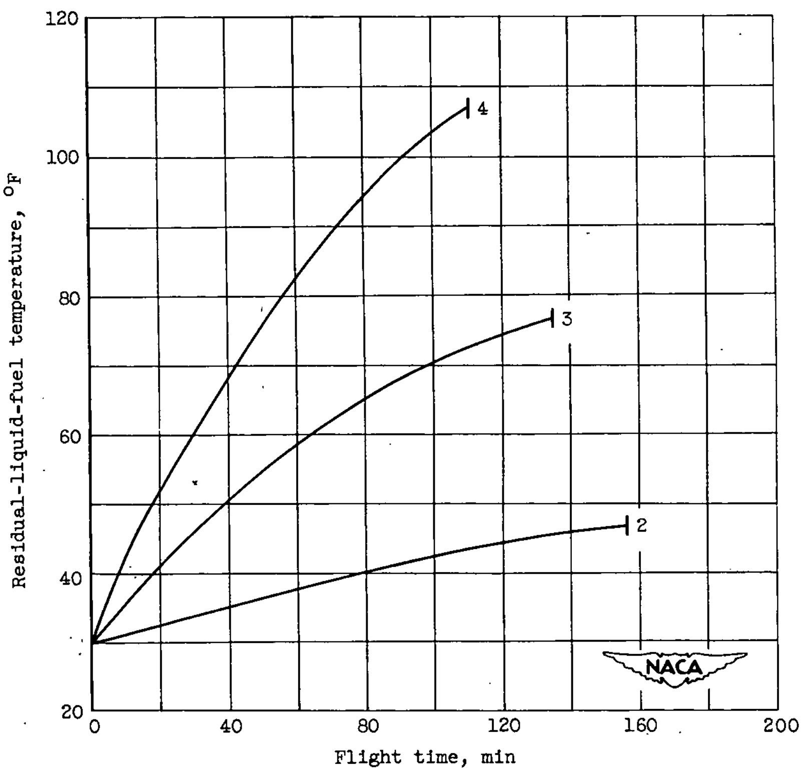


Figure 5. - A.S.T.M. distillation curves for MIL-F-7914 grade JP-5 fuel at two pressure conditions.

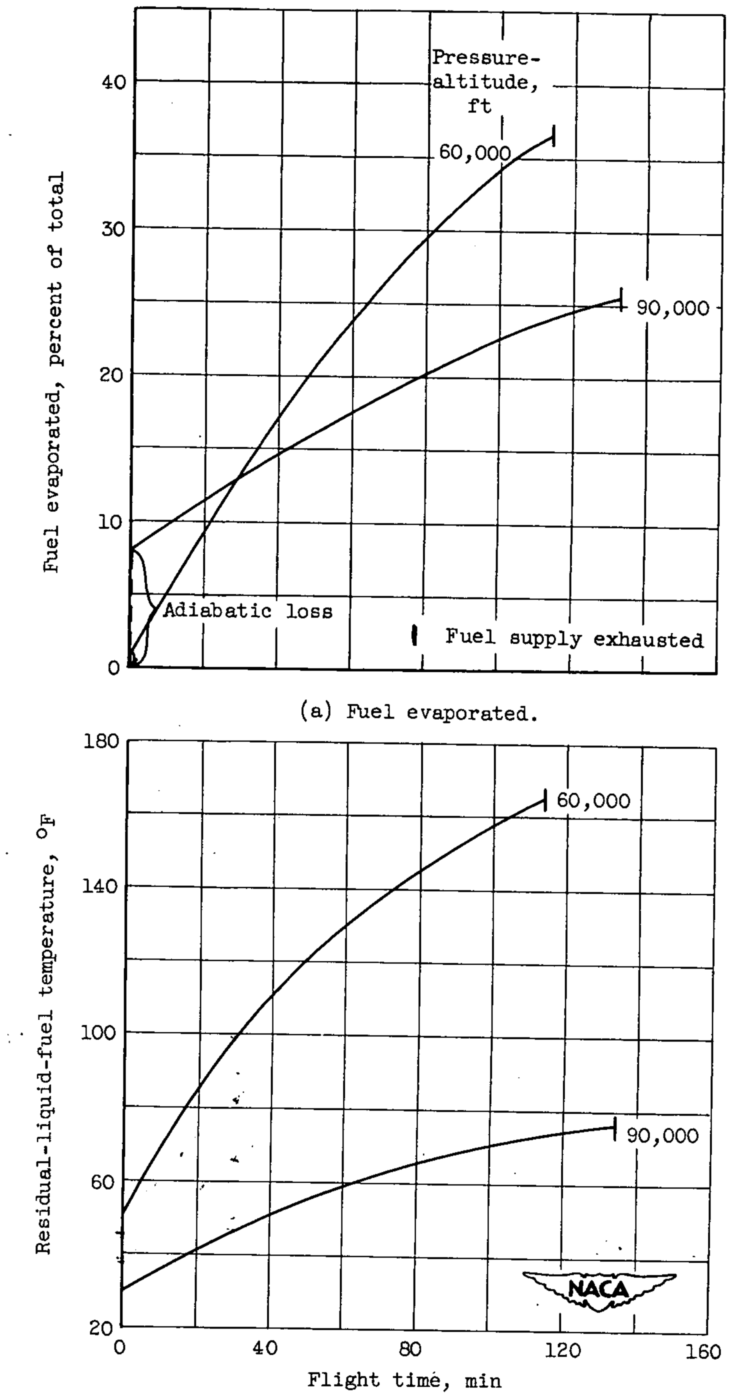


(a) Fuel evaporated.



(b) Residual-liquid-fuel temperature.

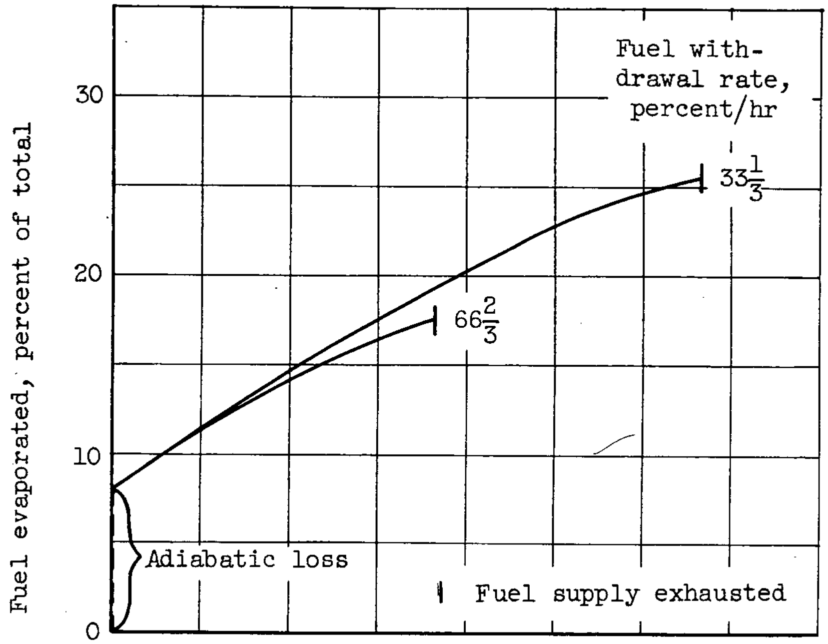
Figure 6. - Effect of flight speed on variation of fuel evaporated and residual-liquid-fuel temperature with flight time for integral uninsulated tank. Altitude, 90,000 feet; initial fuel temperature, 60° F; tank pressurization, 0; tank diameter, 6 feet; fuel, MIL-F-5624A grade JP-4.



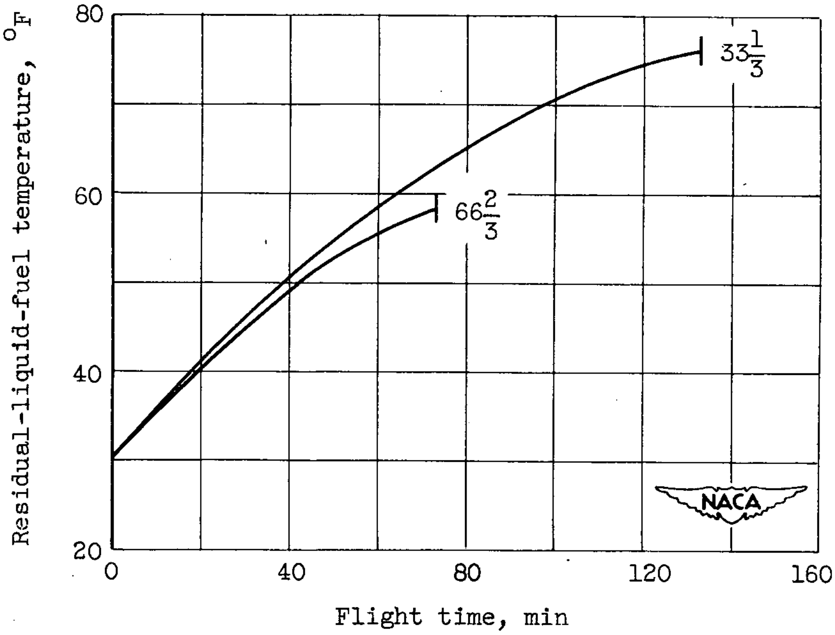
(a) Fuel evaporated.

(b) Residual-liquid-fuel temperature.

Figure 7. - Effect of pressure-altitude on variation of fuel evaporated and residual-liquid-fuel temperature with flight time for integral uninsulated tank. Flight Mach number, 3; initial fuel temperature, 60° F; tank pressurization, 0; tank diameter, 6 feet; fuel, MIL-F-5624A grade JP-4.

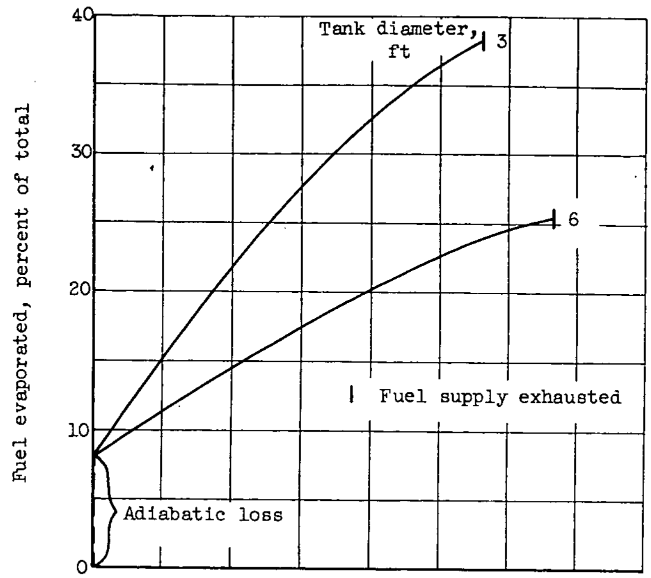


(a) Fuel evaporated.

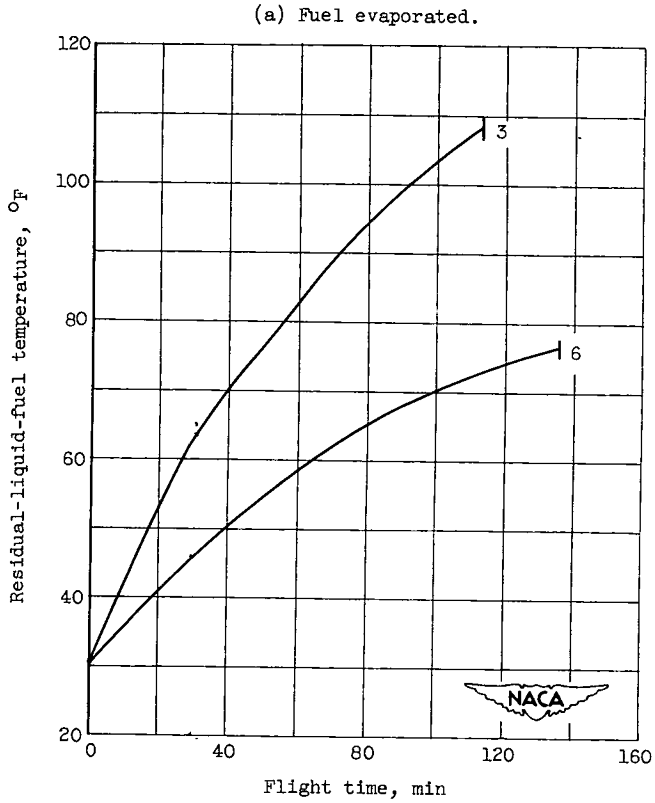


(b) Residual-liquid-fuel temperature.

Figure 8. - Effect of fuel withdrawal rate on variation of fuel evaporated and residual-liquid-fuel temperature with flight time for integral uninsulated tank. Altitude, 90,000 feet; flight Mach number, 3; initial fuel temperature, 60° F; tank pressurization, 0; tank diameter, 6 feet; fuel, MIL-F-5624A grade JP-4.

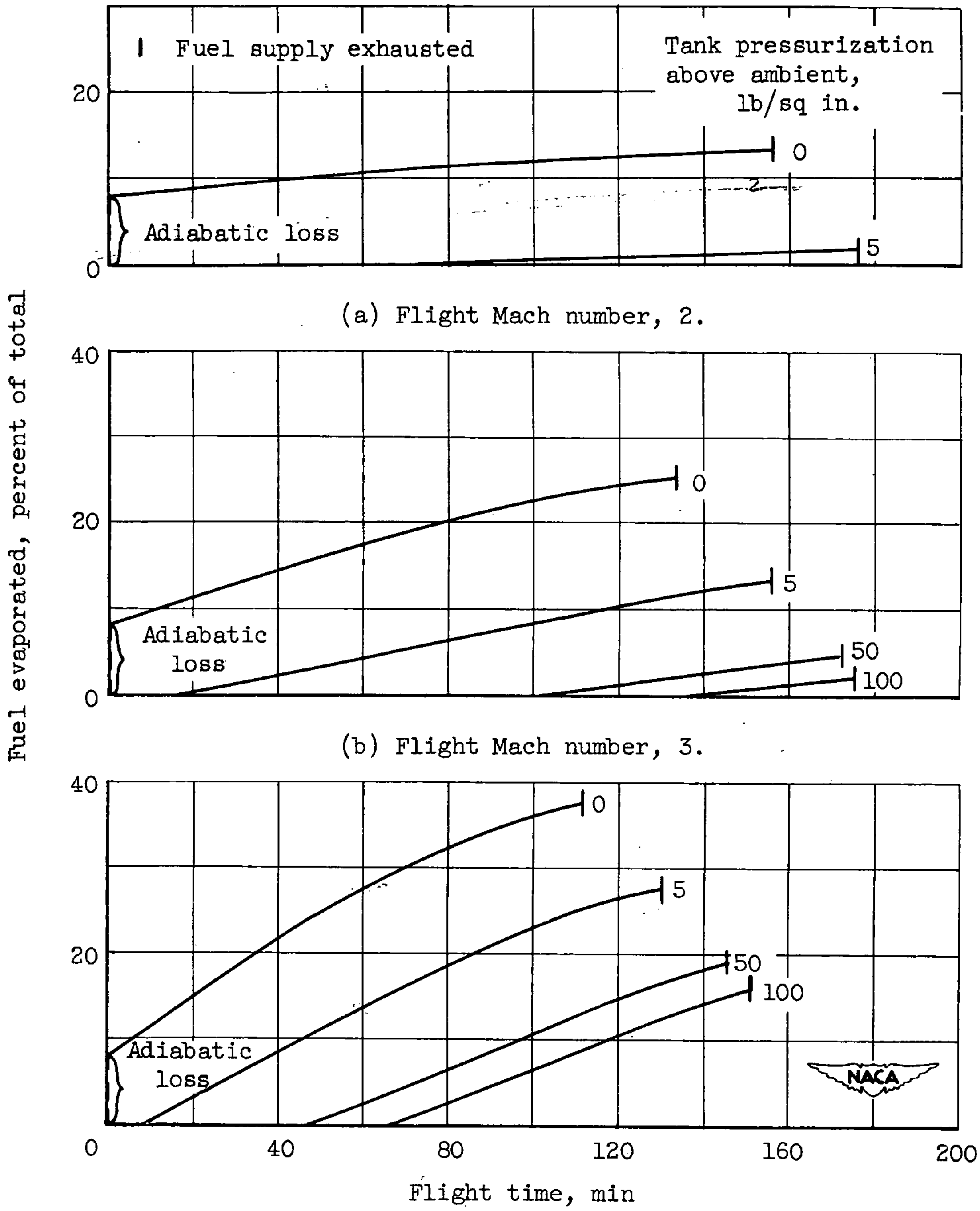


(a) Fuel evaporated.



(b) Residual-liquid-fuel temperature.

Figure 9. - Effect of tank diameter on variation of fuel evaporated and residual liquid-fuel temperature with flight time for uninsulated integral tanks. Altitude, 90,000 feet; flight Mach number, 3; tank pressurization, 0; initial fuel temperature, 60° F; tank length-diameter ratio, 6; fuel, MIL-F-5624A grade JP-4.



(a) Flight Mach number, 2.

(b) Flight Mach number, 3.

(c) Flight Mach number, 4.

Figure 10. - Effect of tank pressurization on variation of fuel evaporated with flight time for integral uninsulated tank at three flight speeds. Altitude, 90,000 feet; initial fuel temperature, 60° F; tank diameter, 6 feet; fuel, MIL-F-5624A grade JP-4.

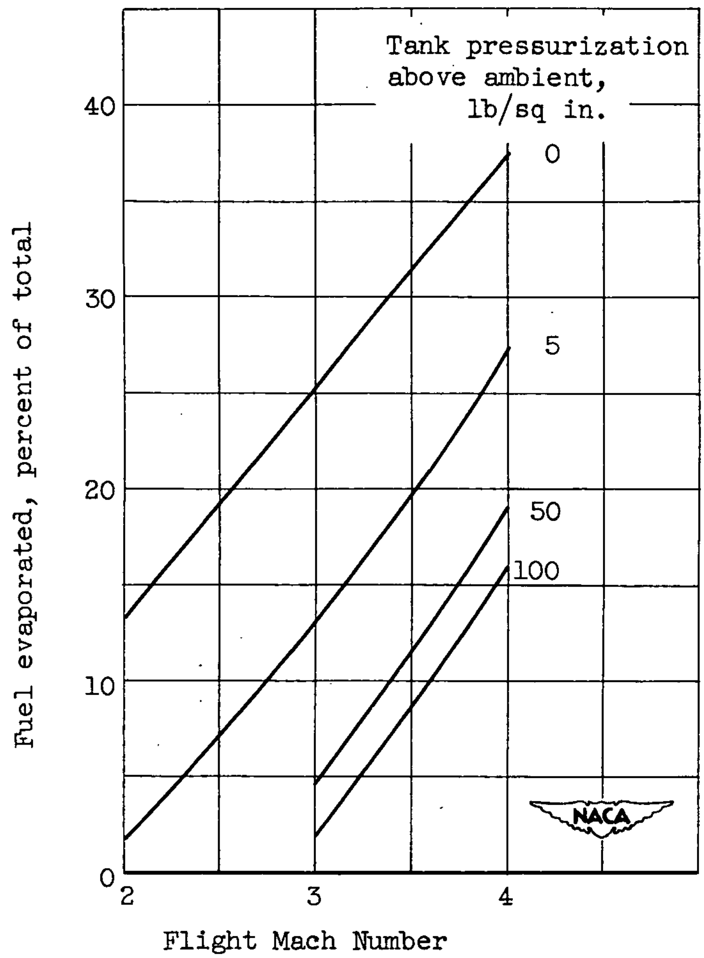


Figure 11. - Variation of fuel evaporated with flight speed for integral uninsulated tank at various tank pressures. Altitude, 90,000 feet; initial fuel temperature, 60° F; tank diameter, 6 feet; fuel, MIL-F-5624A grade JP-4.

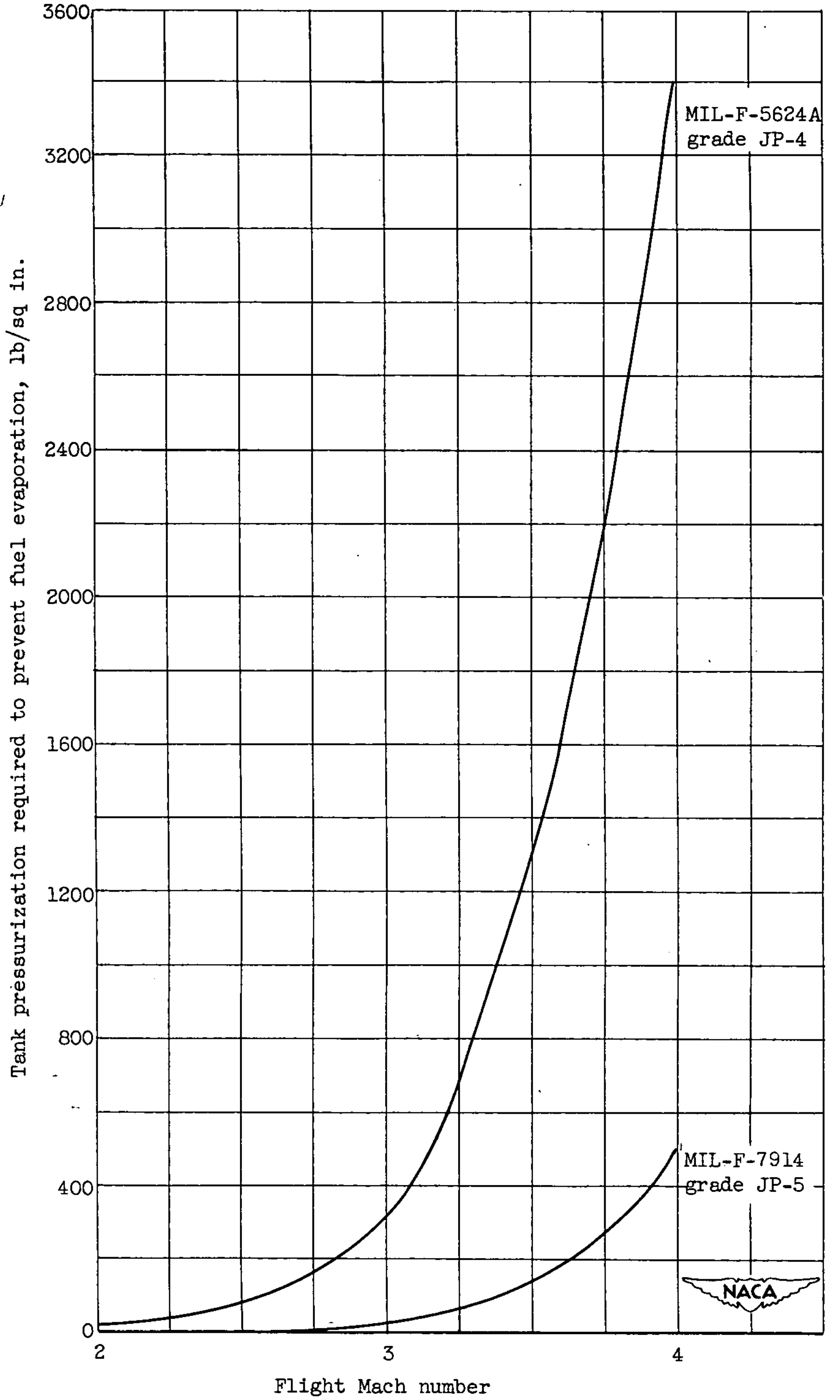


Figure 12. - Variation in tank pressurization required to prevent fuel evaporation in integral uninsulated tank over range of Mach numbers from 2 to 4. Altitude, 90,000 feet; tank diameter, 6 feet; flight time, 3 hours; initial fuel temperature, 60° F; fuels, MIL-F-5624A grade JP-4 and MIL-F-7914 grade JP-5.

[redacted]

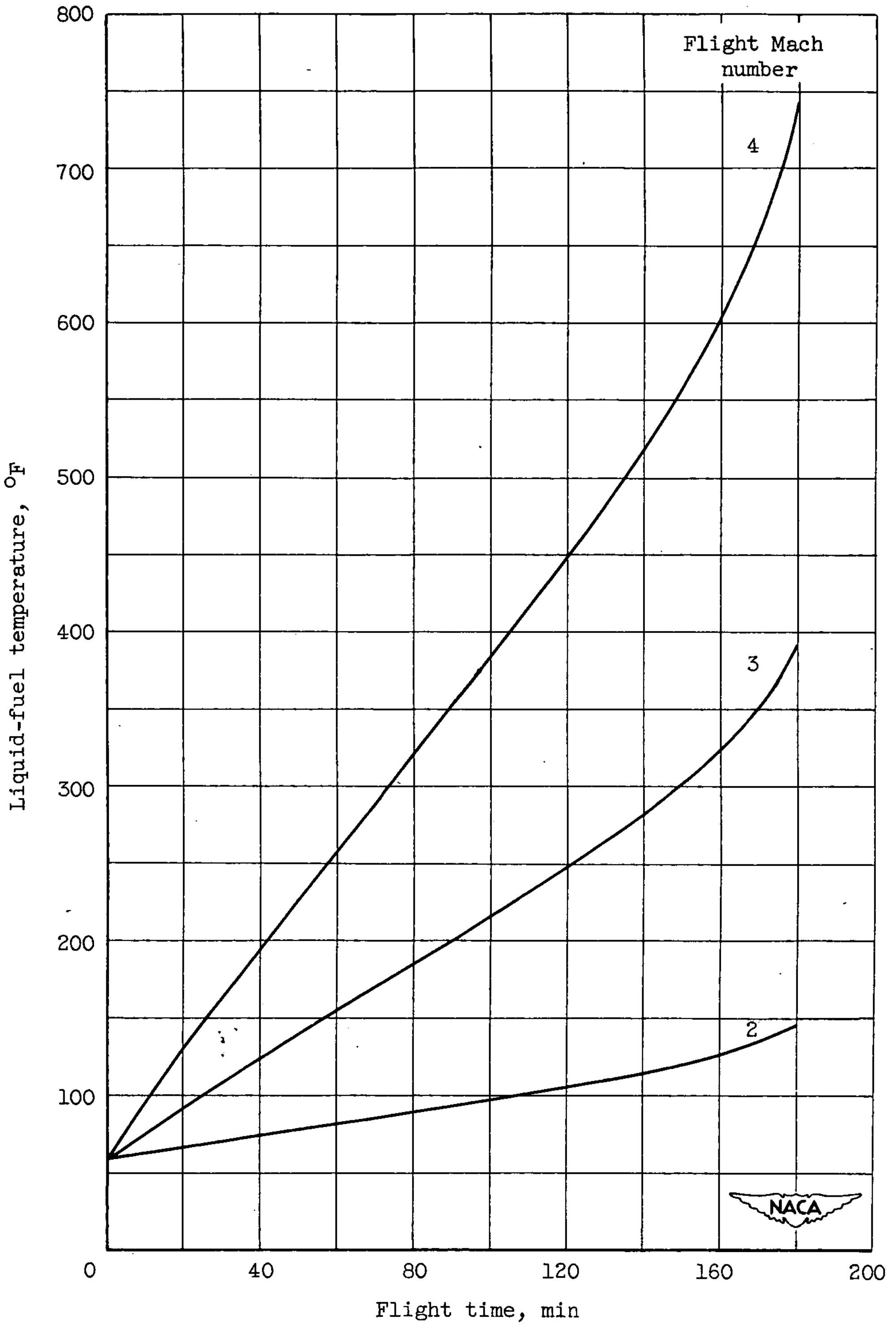


Figure 13. - Effect of flight speed on variation of fuel temperature with flight time in integral uninsulated tank with no fuel evaporation permitted. Altitude, 90,000 feet; initial fuel temperature, 60° F; tank diameter, 6 feet.

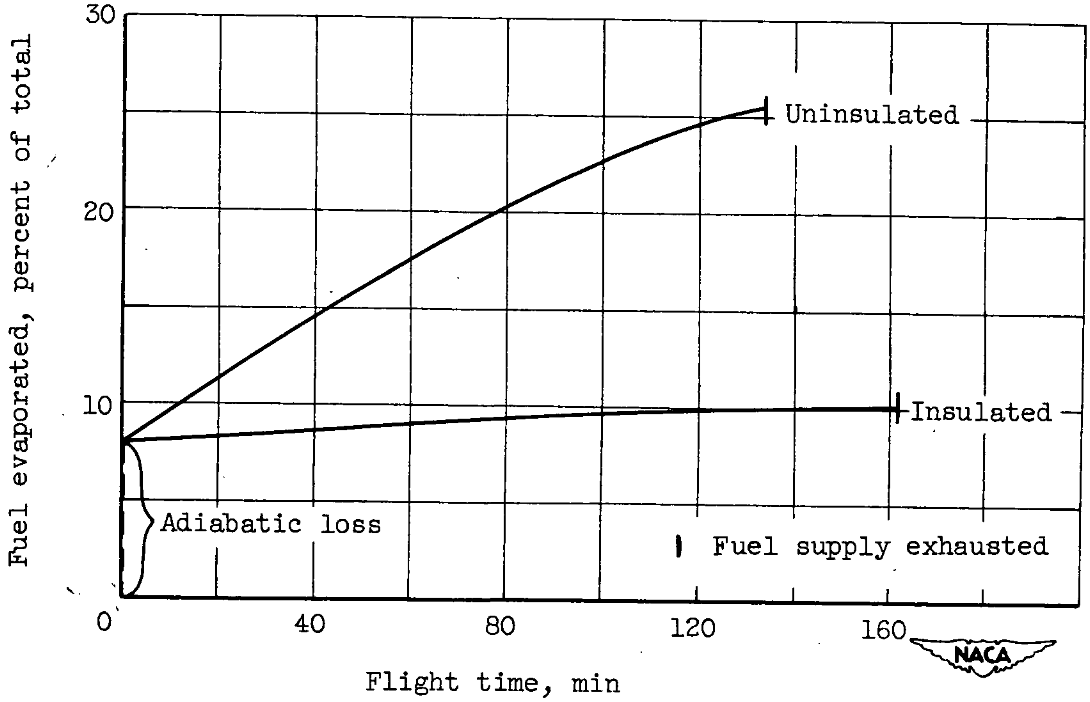
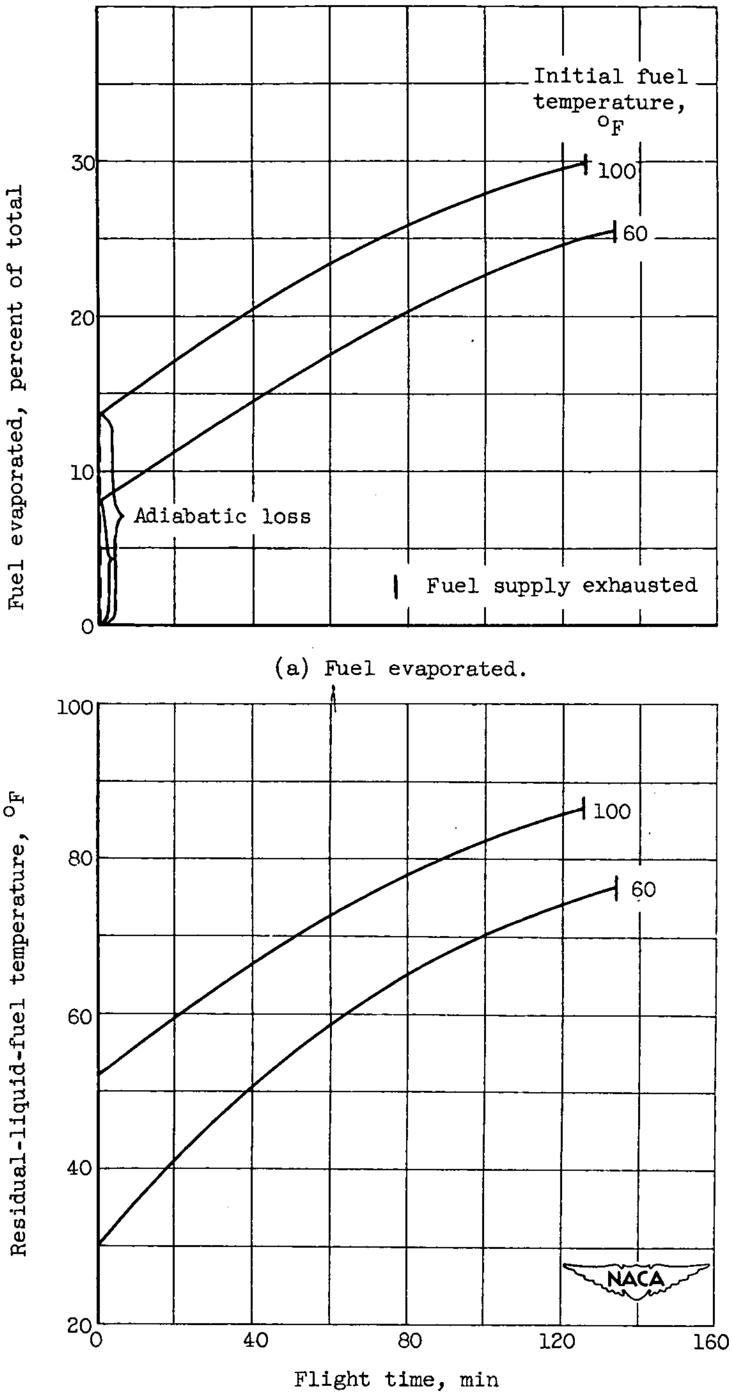


Figure 14. - Effect of tank insulation on variation of percentage of fuel evaporated with flight time. Thermal conductance of insulation, 0.5 Btu/(hr)(ft)^2($^\circ$F); altitude, 90,000 feet; flight Mach number, 3; tank pressurization, 0; initial fuel temperature, 60° F; tank diameter, 6 feet; fuel, MIL-F-5624A grade JP-4.



(a) Fuel evaporated.

(b) Residual-liquid-fuel temperature.

Figure 15. - Effect of initial fuel temperature on variation of fuel evaporated and residual-liquid-fuel temperature with flight time for integral uninsulated tank. Altitude, 90,000 feet; flight Mach number, 3; tank pressurization, 0; tank diameter, 6 feet; fuel, MIL-F-5624A grade JP-4.

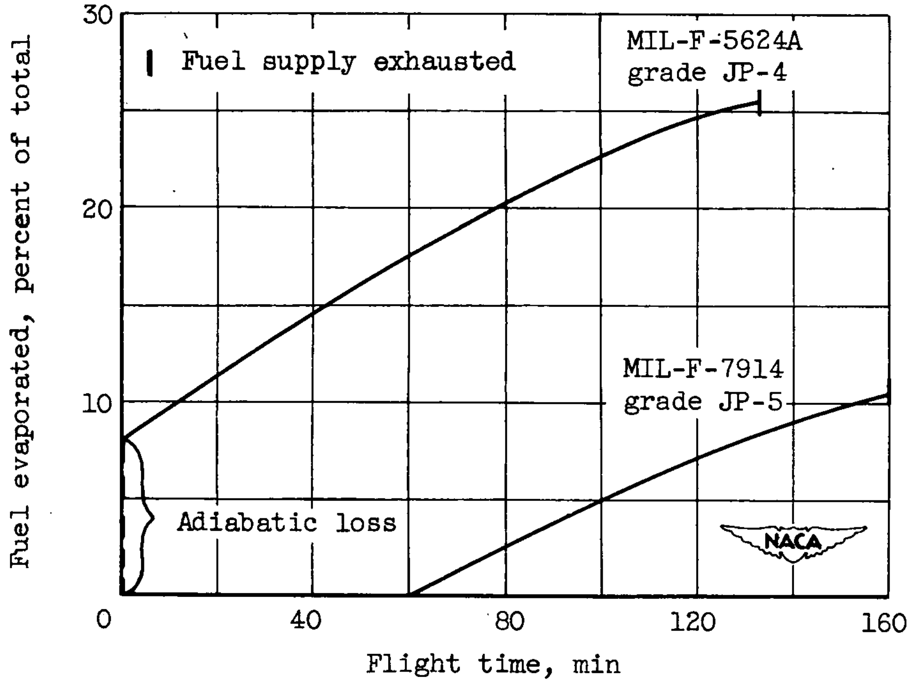


Figure 16. - Effect of fuel volatility on variation of fuel evaporated with flight time for integral uninsulated tank. Altitude, 90,000 feet; flight Mach number, 3; tank pressurization, 0; initial fuel temperature, 60° F; tank diameter, 6 feet.

[redacted]